NASA Contractor Report 3775

Development of MCAERO Wing Design Panel Method With Interactive Graphics Module

J. Dennis Hawk and Dean R. Bristow

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Development of MCAERO Wing Design Panel Method With Interactive Graphics Module

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LIST OF SYMBOLS

AREA	-	Panel Area
c	-	Coefficient in constraint equation
C_p	-	Pressure Coefficient
CRHS	-	Right-Hand-Side value in constraint equation
i	-	Segment column index
j	-	Segment row index
k	-	Index of perturbations
M	-	Mach number
NCON	-	Number of constraint equations
ND	-	Number of perturbations in a constraint equation
NDES	-	Number of prescribed aerodynamic quantities
NKS	-	Number of perturbations
Q	-	Aerodynamic property (velocity or pressure)
q	-	Local velocity on a design panel
WCON	-	Weighting of constraint equation
WDES	-	Weighting of prescribed pressure
V	-	Velocity
(x,y,z)	-	Global Cartesian coordinates
α	-	Angle of attack
β	-	Compressibility factor $(1 - M_\infty{}^2)$
μ	-	Doublet singularity strength
σ	-	Local source singularity strength
ϕ	-	Perturbation potential

Subscripts

A	-	Beginning index
B	-	Ending index
N	-	Normal direction
P	-	Prescribed value
∞	-	Free stream

SUMMARY

A reliable and efficient iterative method has been developed for designing wing section contours corresponding to a prescribed subcritical pressure distribution. The design process is initialized by using MCAERO (MCAIR 3-D Subsonic Potential Flow Analysis Code) to analyze a baseline configuration. A second program DMCAERO is then used to calculate a matrix containing the partial derivative of potential at each control point with respect to each unknown geometry parameter by applying a first-order expansion to the baseline equations in MCAERO. This matrix is calculated only once but is used in each iteration cycle to calculate the geometry perturbation and to analyze the perturbed geometry. The potential on the new geometry is calculated by linear extrapolation from the baseline solution. This extrapolated potential is converted to velocity by numerical differentiation, and velocity is converted to pressure using Bernoulli's equation. There is an interactive graphics option which allows the user to graphically display the results of the design process and to interactively change either the geometry or the prescribed pressure distribution. Not only is this design procedure accurate for large perturbations, but the cost of each iteration cycle is more than two orders of magnitude less than a conventional analysis solution. Examples of the design process are presented to demonstrate that the method is accurate, numerically stable, and converges in only three to five iterations.

INTRODUCTION

The surface singularity panel method has been widely accepted as an excellent means of determining the subcritical, potential flow about complex aircraft configurations (References 1-9). The better formulated methods will consistently predict accurate wing pressure distributions, even in regions with strong fuselage-nacelle-store interfernce. A typical example of this accuracy obtained by MCAERO (MCAIR 3-D Subsonic Potential Flow Analysis Code) is shown in Figure 1 for an F-18 configuration. The power of surface panel methods has been recognized by several investigators who have developed iterative inverse techniques for designing wing design section geometry corresponding to prescribed pressure distributions (References 10-13). However, each of these investigators have encountered and not overcome at least two of the shortcomings listed in Figure 2.

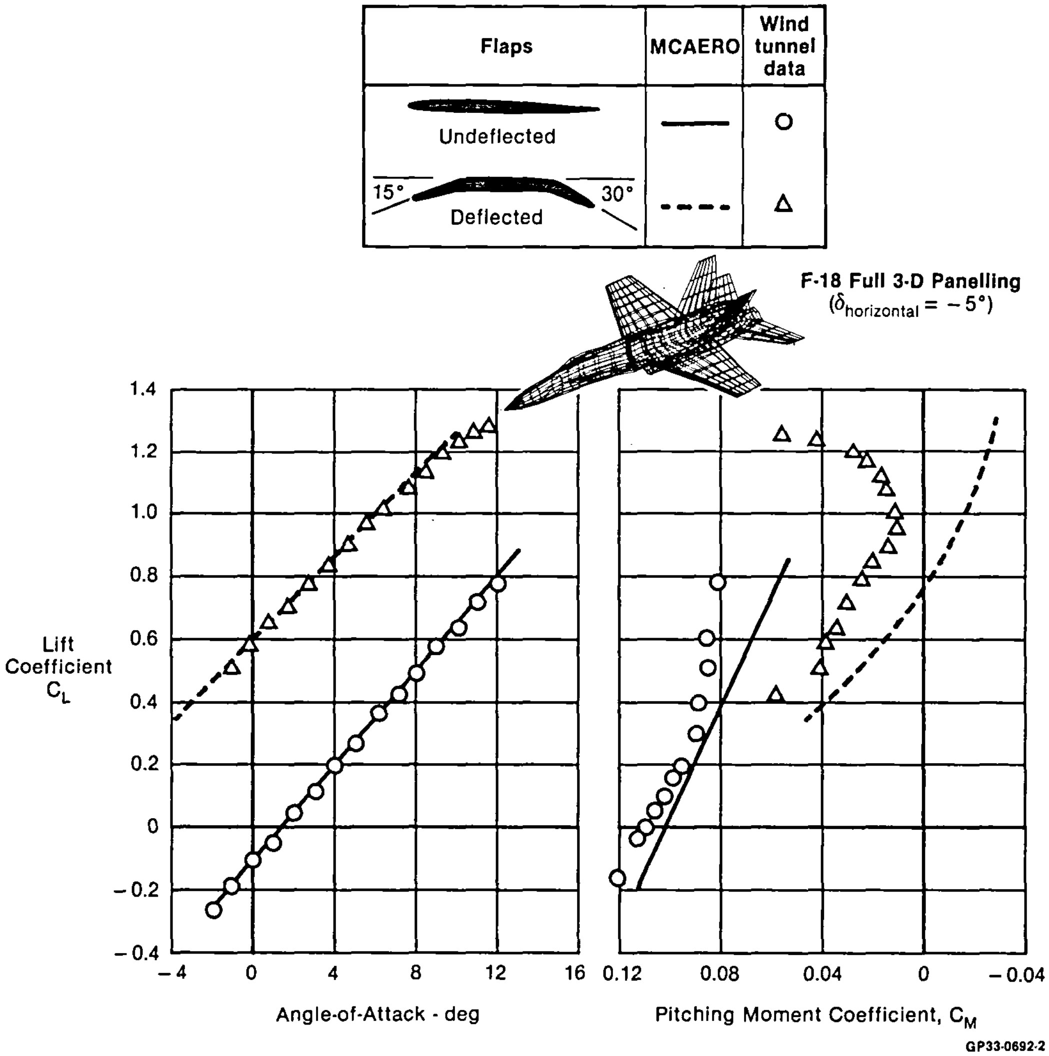

Figure 1. MCAERO Prediction Accuracy for Clean and Takeoff Configurations

Symptoms	Cause	Remedy
• Calculated Pressures Overly Sensitive to Contour Smoothness	• Source Singularities	• Combined Source - Doublets (Green's Identity)
• Calculations Do Not Converge • Inaccurate Leading Edge Design Contours	• Dirichlet Boundary Conditions • Transpiration Principal (Equivalent Blowing)	• First-Order Mathematical Expansion ($\partial C_p/\partial z$)
• Unrealistically Wavy Design Contours	• One Unknown Coordinate per Prescribed Pressure	• More Pressures Than Unknowns • Least Squares
• High Cost per Iteration	• Complete Panel Method Analysis During Iteration Cycle	• Perturbation Analysis Method

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Figure 2. Common Barriers to a Successful Wing Design Method

A reliable and extremely efficient method for solving the wing-on-fuselage design problem depicted in Figure 3 was developed in 1982 (Reference 14). The development of this method by McDonnell Aircraft Company (MCAIR) was supported under contract to NASA Langley Research Center (LaRC). The efficiency is the result of employing the pertubation analysis method of References 8 and 9 in each iteration cycle of the design solution. The design approach is based upon the execution of three complementary but independent computer programs. For a given baseline panelled configuration, the first program generates a conventional panel method analysis solution, including surface singularity strengths and pressure distribution. The solution file from the first program provides the input to the second program, which calculates a matrix of partial derivatives of surface potential with respect to arbitrary geometry perturbations. The first and second program are executed one time only in order to provide a permanent input file to the third program, which calculates the wing geometry corresponding to a prescribed pressure distribution.

As part of the MCAIR 1982 and 1983 Independent Research and Development Project, user-oriented production versions of the above three computer programs were developed. These programs are respectively designated "MCAERO", "DMCAERO", and "DESIGN". A fourth production program that applies the perturbation analysis method of References 8 and 9 is designated "PAM".

As part of the present contract with LaRC (NAS1-17176), an interactive graphics option for the design method was established. This report summarizes the capabilities of the user-oriented production codes and the interactive graphics design option.

Given:
- Fuselage geometry
- Baseline wing
- Angle-of-attack
- Prescribed pressure distribution (C_{p_i})

Calculate: Wing section geometry

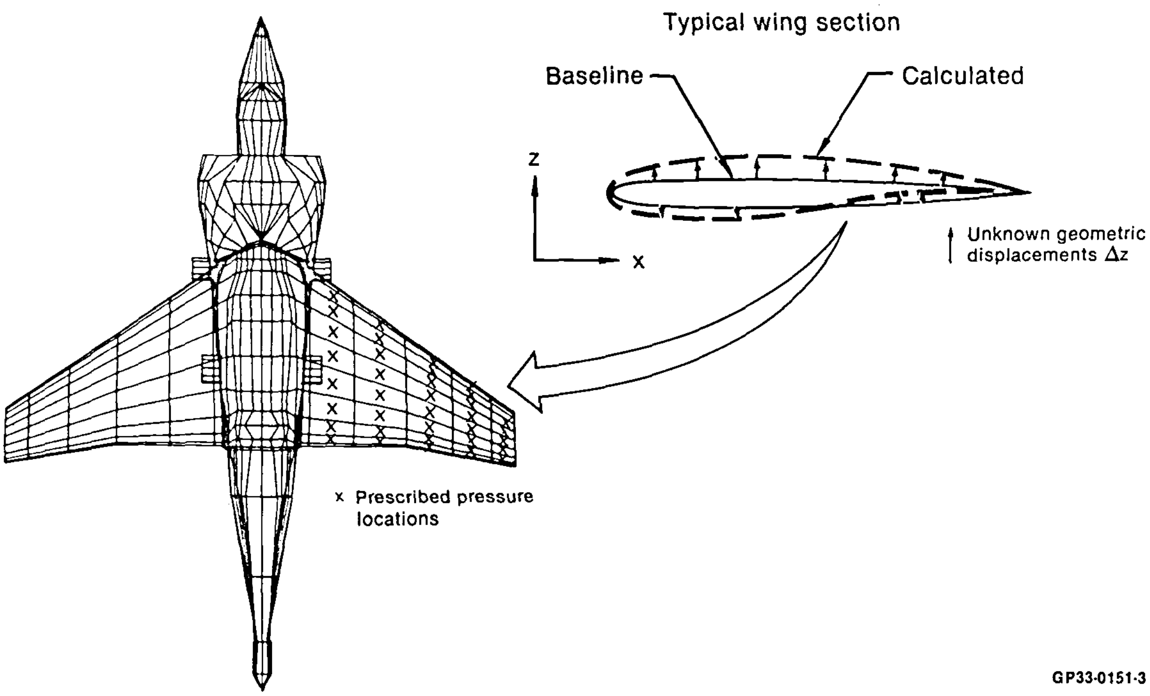

Figure 3. Objective of the Design Method

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2. PERTURBATION ANALYSIS METHOD

The standard solution approach to prescribed pressure design problems is to divide each iteration cycle into an analysis, pressure calculation step and an inverse, geometry correction step. In the two-dimensional method of references 7 and 15, an entire panel method solution is calculated during each analysis step. Furthermore, a new geometry-velocity perturbation matrix is calculated for each inverse step. In spite of the fact that the number of computations in each iteration cycle is a cubic function of the number of panels, the total computing cost is relatively small. The reason is that with the better formulated methods typical two-dimensional problems require fewer than one hundred panels.

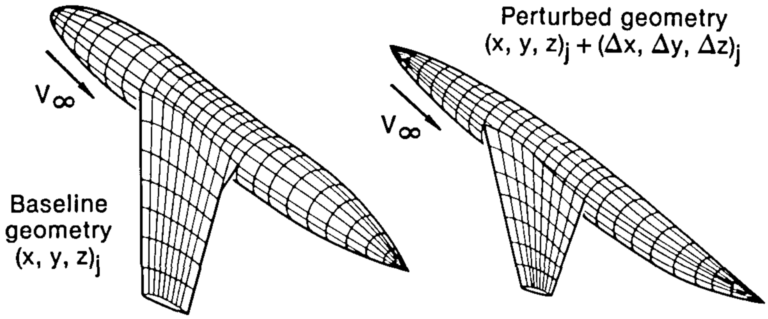
However, the number of panels required for wing-fuselage configurations is almost an order of magnitude greater, and the approach of the two-dimensional design procedure would be extremely expensive. During the initial development of the pilot code version of the wing design method, it became evident that the cost of a practical wing-on-fuselage design procedure would be prohibitive using the then existing panel method technology. Therefore, in order for the design procedure to become practical, the existing panel method technology was expanded by a new cost savings feature, i.e., the Perturbation Analysis Method (references 8 and 9). The method is an extremely efficient tool for analyzing the pressure distribution corresponding to a series of arbitrary, small perturbations to a baseline wing-fuselage geometry (Figure 4). The following features of the perturbation analysis method make it especially practical for application to an iterative wing section design method.

(1) The computational expense for analyzing each successive geometry perturbation is almost two orders of magnitude less than that of a conventional panel method,

(2) The pressure distribution prediction accuracy is competitive with conventional surface panel methods for very large perturbations to wing section geometry.

(3) A pre-calculated matrix of partial derivatives for the paneled baseline configuration is available. Each element of the matrix is the rate of change of potential at a boundary condition control point with respect to a geometry parameter perturbation. For design applications, the geometry-potential perturbation matrix can be efficiently converted to a geometry-pressure perturbation matrix.

The purpose of this section is to give a brief review of the fundamentals and power of the Perturbation Analysis Method (a more complete description is available in references 8 and 9).



Objective

- Subsonic inviscid analysis of multiple geometry perturbations at small additional cost

Approach

- Precalculated baseline matrix of potential derivatives $\left\{ \frac{\partial \phi_i}{\partial x_j}, \frac{\partial \phi_i}{\partial y_j}, \frac{\partial \phi_i}{\partial z_j} \right\}$
- Linear extrapolation

$$(\phi_i + \Delta\phi_i) = \phi_i + \sum_j \left\{ \frac{\partial \phi_i}{\partial x_j} \Delta x_j + \frac{\partial \phi_i}{\partial y_j} \Delta y_j + \frac{\partial \phi_i}{\partial z_j} \Delta z_j \right\}$$

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Figure 4. Perturbation Analysis Method

2.1 MATHEMATICAL APPROACH - The perturbation analysis approach requires an initial baseline calculation of a conventional panel method solution for an arbitrary baseline configuration. Subsequently, a matrix consisting of the analytically derived partial derivatives of velocity potential with respect to geometry coordinates is calculated. The baseline solution and derivative matrix are calculated one time only and then stored for repetitive use. For each geometry perturbation, the solution surface distribution of velocity potential is constructed by multiplying the derivative matrix by a new right-hand-side. This procedure bypasses the two computationally expensive steps of a conventional panel method: calculating the influence coefficients and solving a large system of linear algebraic equations.

Although the perturbation analysis method is appropriate for predicting the effect of arbitrary small changes to wing planform and fuselage geometry, the real power of the method is the accuracy with which large perturbations to wing thickness, camber, and twist can be analyzed. Examples of the accuracy and efficiency of this method can be found in references 8, 9, and 14, and therefore will not be repeated here.

The first three computer codes listed in Figure 5 are required in order to use the production version of the Perturbation Analysis Method. The first program, MCAERO, is the MCAIR conventional panel method for analyzing the baseline configuration. The second program DMCAERO, employs a differential mathematical formulation and an output file from MCAERO to calculate the matrix of partial derivatives of the perturbation potential. For each baseline configuration, DMCAERO creates an input file for the third program - Perturbation Analysis Method (PAM). The pilot code versions of both MCAERO and DMCAERO were developed basically to validate the procedure. While both worked successfully, each suffer from a similar problem, tedious input. Therefore a prime concern in the development of the production version of the codes was to simplify the input while improving efficiency and accuracy. Figure 6 shows the success obtained in simplifying the input for both MCAERO and DMCAERO.

	MCAIR 3-D Subsonic Potential Flow Program (MCAERO)	Geometry Influence Coefficient Program (DMCAERO)	MCAIR 3-D Perturbation Analysis Program (PAM)	MCAIR 3-D Wing Design Program (DESIGN)
Input	• Baseline Geometry $(x, y, z)_j$	• Baseline Geometry $(x, y, z)_j$ • Baseline Potential Distribution ϕ_i	• Baseline Properties $(x, y, z)_j$, ϕ_i • Derivative Matrix $(\partial\phi_i/\partial x_j, \partial\phi_i/\partial y_j, \partial\phi_i/\partial z_j)$ • Geometry Perturbation $(\Delta x, \Delta y, \Delta z)_j$	• Baseline Properties $(x, y, z)_j$ ϕ_i • Derivative Matrix $(\partial\phi_i/\partial z_j)$ • Prescribed Pressures
Approach	• Conventional Panel Method	• First-Order Expansion to Panel Method Formulation	• Linear Extrapolation $\Delta\phi_i = \sum_j [(\partial\phi_i/\partial x_j)\,\Delta x_j + (\partial\phi_i/\partial y_j)\,\Delta y_j + (\partial\phi_i/\partial z_j)\,\Delta z_j]$	• Linear Extrapolation and Least Squares Solution
Output	• Baseline Potential Distribution ϕ_i • Baseline Aerodynamic Properties	• Derivative Matrix $(\partial\phi_i/\partial x_j, \partial\phi_i/\partial y_j, \partial\phi_i/\partial z_j)$	• Aerodynamic Properties of Perturbed Geometry	• Designed Wing Geometry

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Figure 5. The Computer Programs for the 3-D Wing Design Method

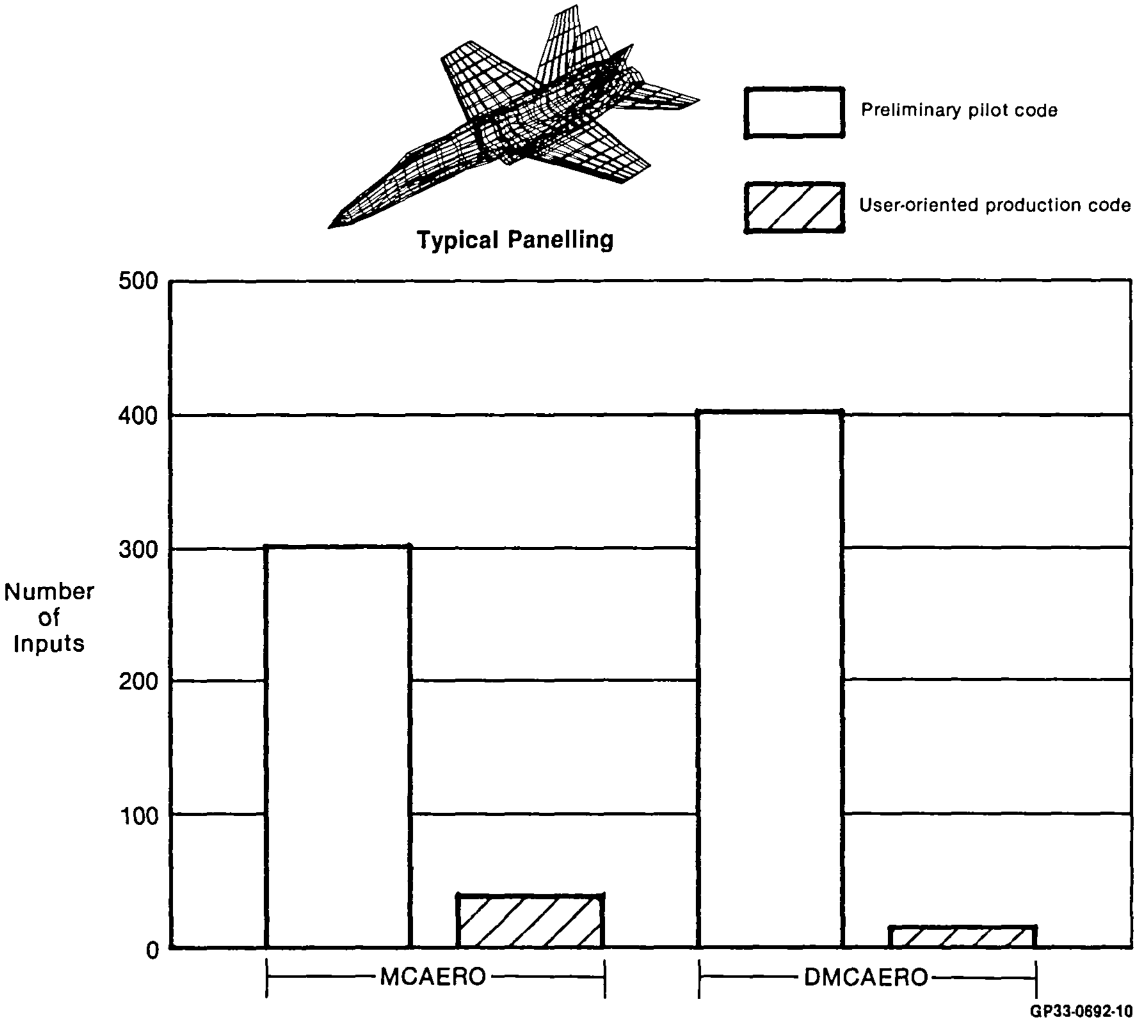
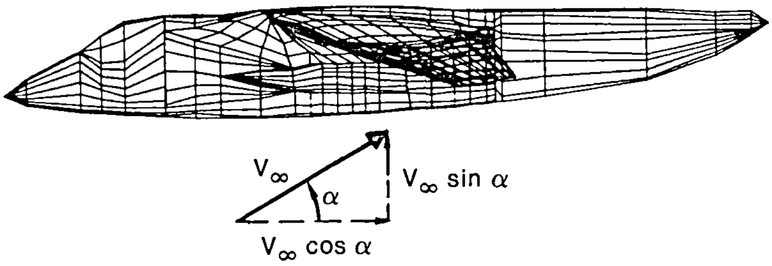


Figure 6. Number of Inputs for a Typical Case

While MCAERO and DMCAERO can be relatively expensive to run, once the input file for PAM has been generated, the first two programs are no longer required. PAM can be executed repeatedly at low computing cost for the analyses of a series of perturbations to the panel corner coordinates $(x,y,z)_j$ of the complete aircraft configuration. The method used to analyze each perturbation is the same as the conventional panel method calculation with two significant exceptions. First, no influence coefficients are calculated; second, no large system of linear algebraic boundary condition equations is solved. Instead, the perturbation potential at each control point is calculated by linear extrapolation. The conversion of potential to surface velocity is based on numerical differentiation.

Usually the second program is executed twice for each baseline configuration, once at 0° incidence and once at 90° incidence. By employing the principle of linear superposition, the perturbation analysis is automatically performed at any intermediate angle of attack (Figure 7).



$$\phi_i = \left\{ \begin{array}{l} \cos \alpha \cdot \left[\phi_{i_{\text{baseline}}} + \sum_j \dfrac{\partial \phi_i}{\partial z_j} \Delta z_j \right]_{0^\circ \alpha} \\ + \sin \alpha \cdot \left[\phi_{i_{\text{baseline}}} + \sum_j \dfrac{\partial \phi_i}{\partial z_j} \Delta z_j \right]_{90^\circ \alpha} \end{array} \right\}$$

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Figure 7. Linear Superposition of 0° and 90° Solutions
3-D Wing Design Method

Figure 8 shows the relative cost of each of the codes listed in Figure 5 for a typical wing only and wing-on-fuselage case. As can be seen once the "expensive" codes (MCAERO and DMCAERO) are run, the computing costs become insignificant.

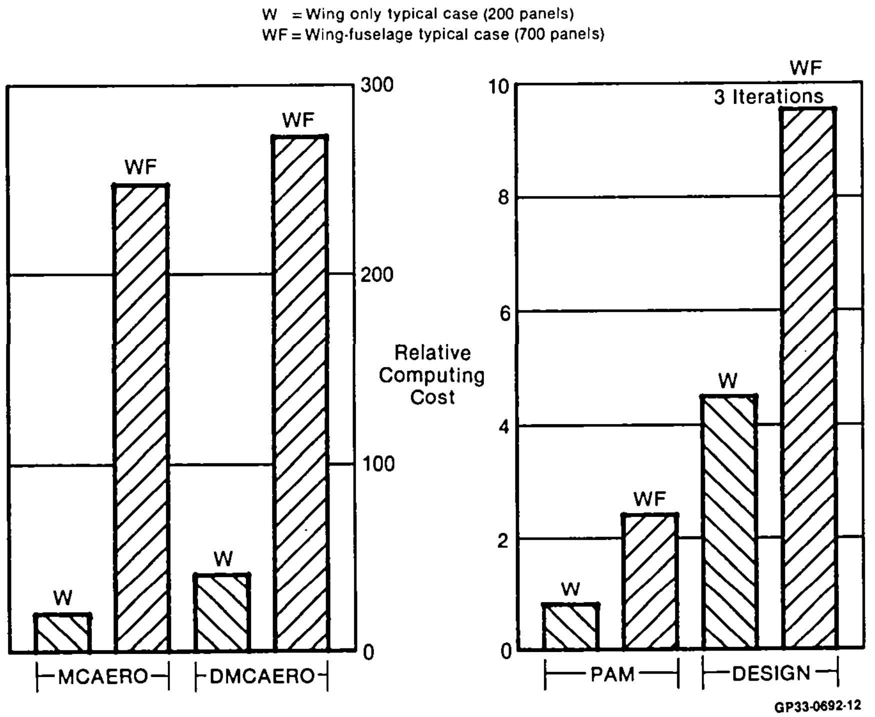


Figure 8. Relative Computing Costs for the Four Codes Used in the Wing Design Method

3. 3-D WING DESIGN METHOD

For a given panelled wing or wing-fuselage configuration, assume that prescribed pressure coefficients for a fixed arbitrary angle of attack are assigned to the panel centers of the wing. The objective of the design method is to determine the change in wing section geometry that most nearly corresponds to the prescribed pressure distribution (Figure 3).

As was shown previously, the Perturbation Analysis Method can accurately calculate the pressure distribution corresponding to large changes in wing section geometry. With that as a basis, it was reasoned that an inverse formulation to the Perturbation Analysis Method could accurately design large changes in wing section geometry. It was this approach that formed the basis of the wing design method. The design method must employ an iterative scheme because pressure is a nonlinear function with respect to geometric changes. However, recognition of the fact that the perturbation potential is a nearly linear function eliminated the need to perform extensive computations in each iteration cycle.

A schematic of the wing design method is presented in Figure 9. The method can be used to solve very general aerodynamic design problems. For example, the prescribed aerodynamic quantity at a panel center can be either a velocity component or pressure coefficient. Arbitrary geometry parameters such as wing chord or fuselage shape can be selected for design. Furthermore, design constraints such as fixed camber or thickness can be imposed. Most design problems, however, are of the type illustrated in Figure 3. This type, designated the "standard wing design problem", is defined in detail below. The remainder of this section presents the mathematical formulation for the wing design method and example design solutions.

3.1 <u>STANDARD WING DESIGN PROBLEM</u> - As illustrated in Figure 10, the region of panels subject to design is identified by corner points in the range $(i_A, j_A) \leq (i,j) \leq (i_B, j_B)$, where the limits (i_A, i_B, j_A, j_B) are selected by the user. If j_A and j_B are points on the upper and lower surface trailing edge respectively, then the geometry of the complete wing section at each span station i will be designed. One restriction of the wing panelling is that it be basically trapezoidal. At the center of each panel in the design region, the desired pressure coefficient is prescribed by the user.

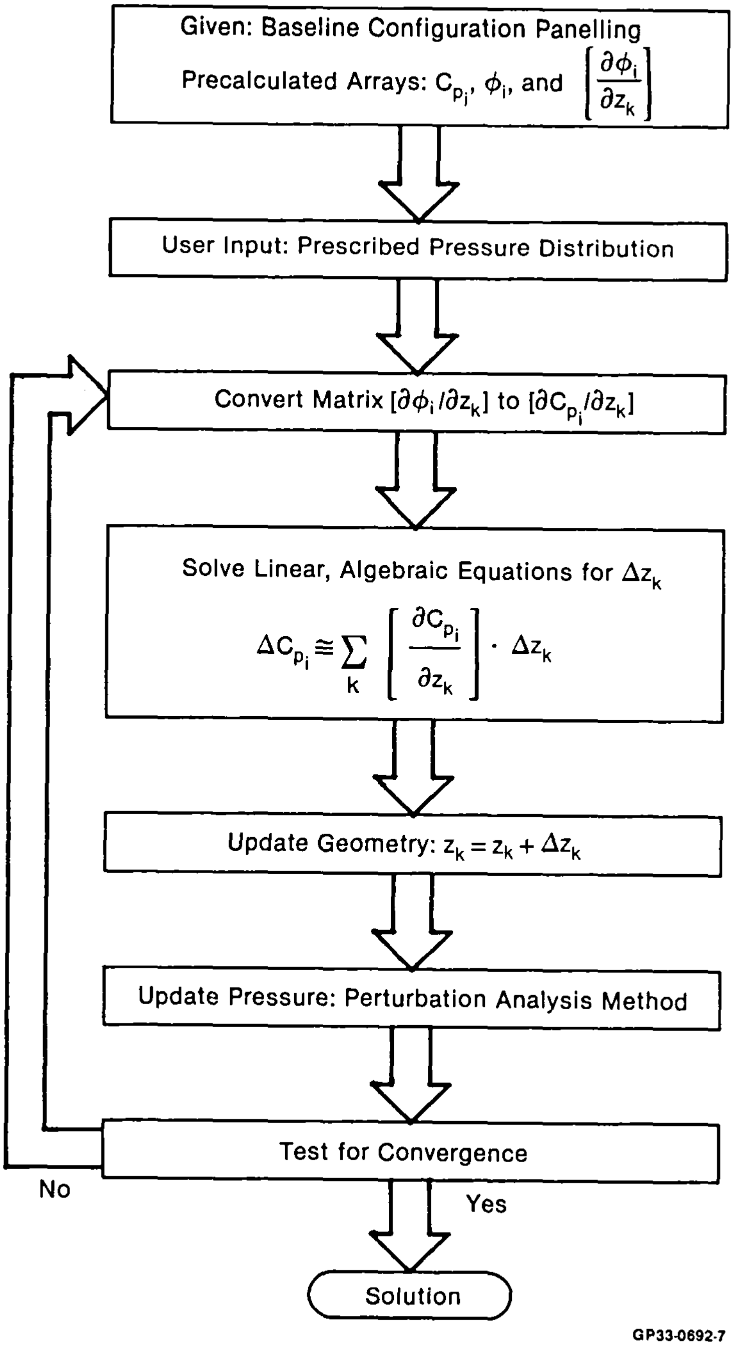


Figure 9. Schematic of 3-D Wing Design Method

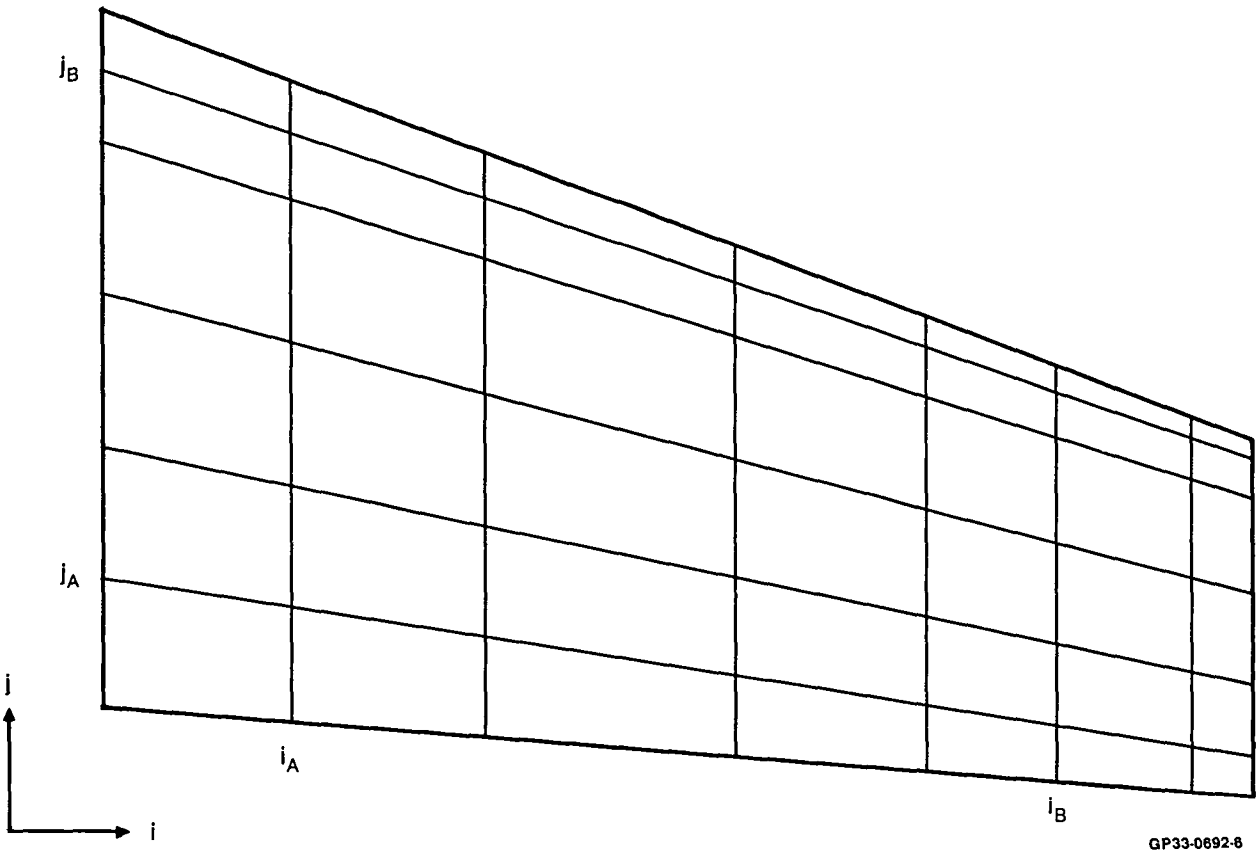


Figure 10. Projection of a Design in x-y Region

The unknowns are Δz at the panel corner points $\{(i_A, j_A+1) \leq (i,j) \leq (i_B, j_B-1)\}$. However, not all of the unknowns are permitted to be independent. As illustrated in Figure 11, less than one-half of the unknowns are independent. The remaining unknowns are generated by interpolation through the independent unknowns. On the span stations $i = i_A, i_A+2, i_A+4, \ldots, i_B$, each dependent unknown $\Delta z(i,j)$ is established by least squares quadratic interpolation through the path of points $(j-3, j-1, j, j+1, j+3)$ on the baseline configuration. For the remaining span stations ($i = i_A+1, i_A+3, \ldots, i_B-1$), each unknown $\Delta z(i,j)$ is established by straight line interpolation through $\Delta z(i-1,j)$ and $\Delta z(i+1,j)$. For this type of interpolation to be accurate, it is implicitly assumed that the three points $(i-1,j)$, (i,j), and $(i+1,j)$ lie on nearly the same per cent chord line. Typical wing panelling is consistent with this assumption.

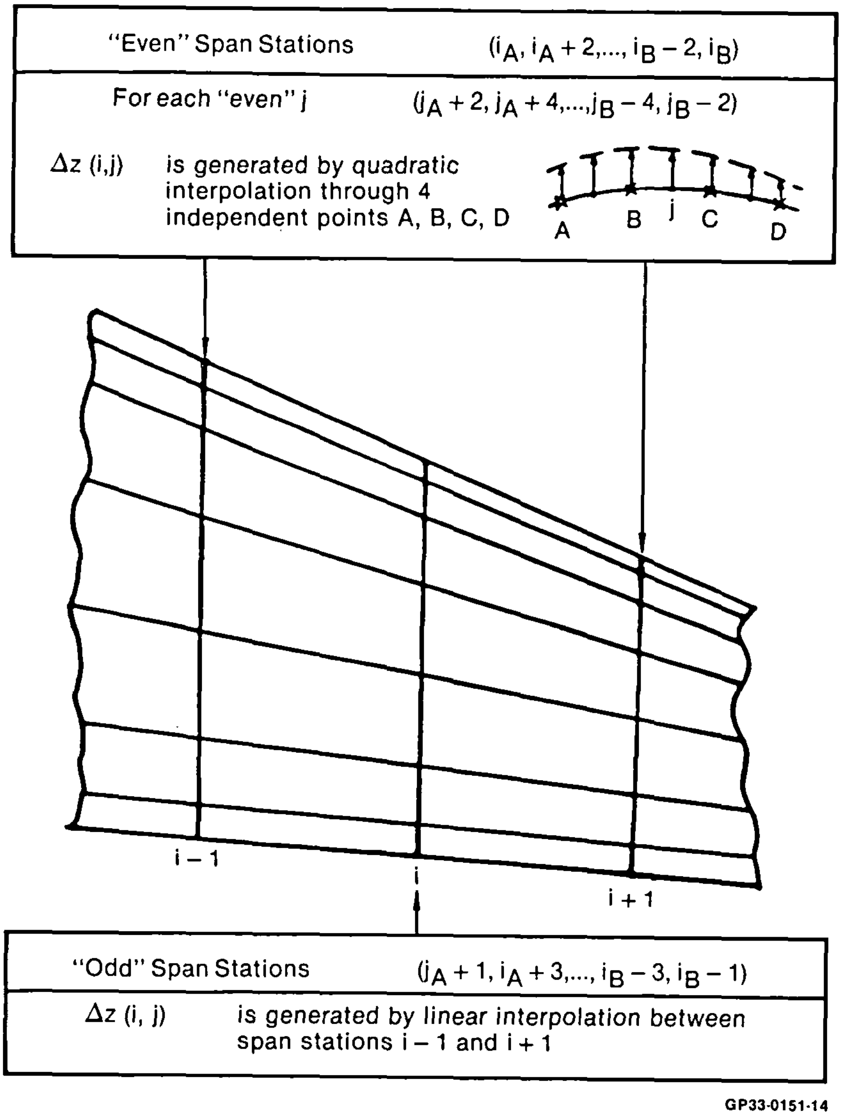


Figure 11. Interpolation Scheme for Dependent Unknowns

The reason for limiting the number of independent unknowns a priori is to prevent numerically unstable design calculations. Figures 12a and 12b illustrate two types of design instabilities that could occur if every panel corner in the design region were allowed to be an independent unknown.

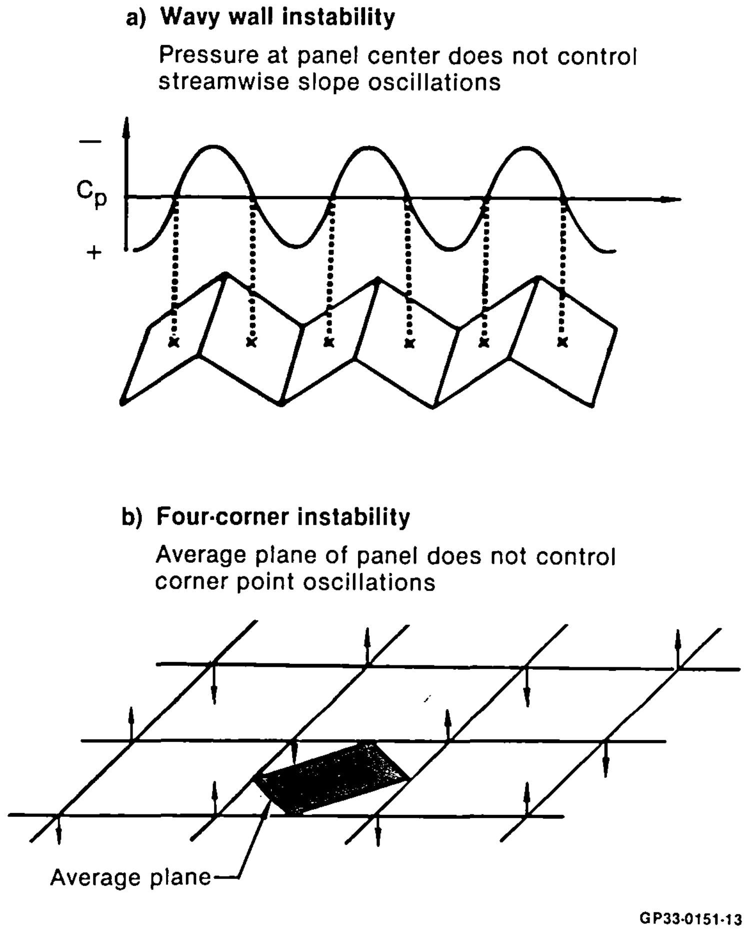


Figure 12. Typical Wing Design Solution Instabilities

Consistent with the nomenclature of the Perturbation Analysis Method (reference 8), each independent unknown perturbation is assigned an index k_S. The value of Δz for perturbation number k_S is designated S_{k_S}. A schematic of the independent unknowns is presented in Figure 13. The objective of the wing design method is to calculate the values S_{k_S} ($1 \leq k_S \leq NKS$) such that the resulting pressure distribution most nearly corresponds to the prescribed pressure distribution.

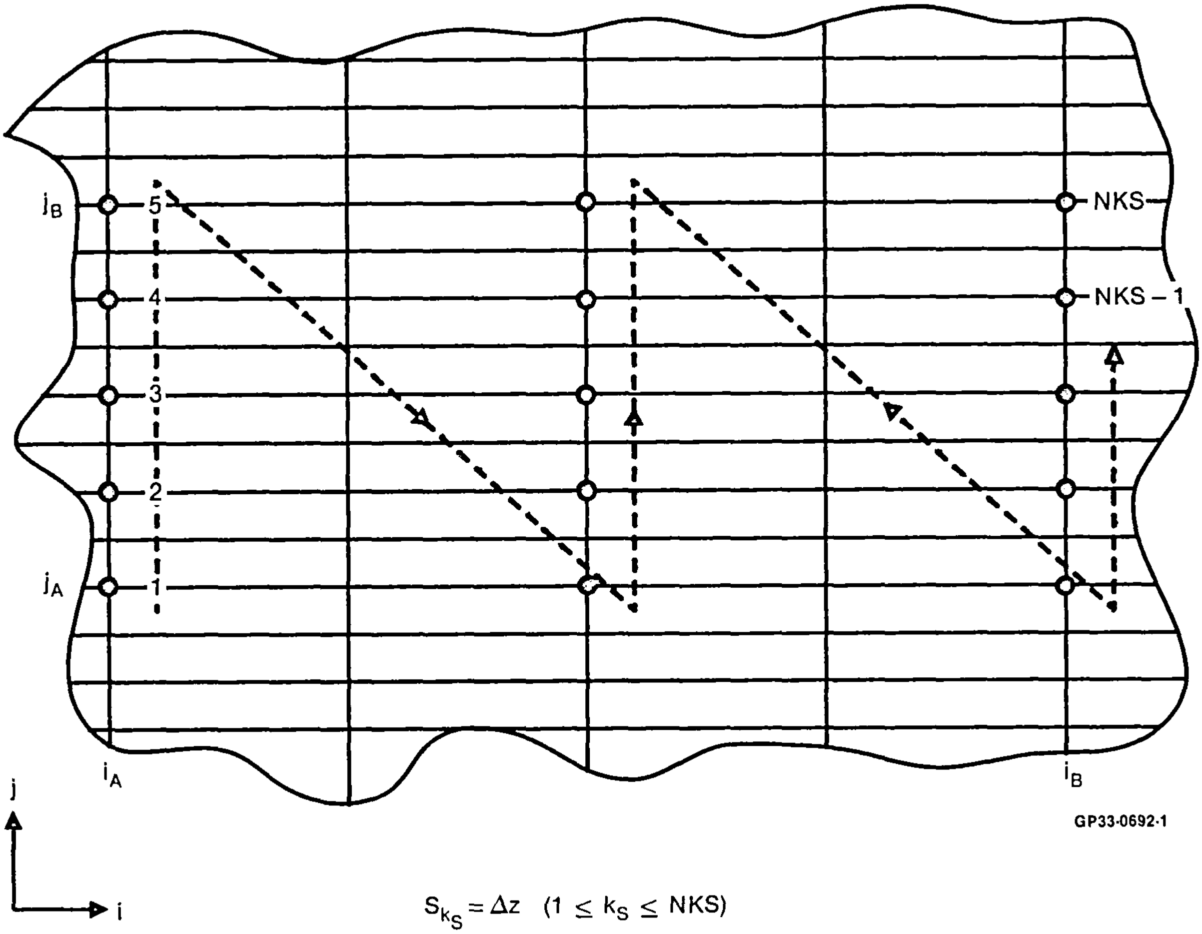


Figure 13. Ordering of Independent Unknowns S_{k_S}

3.2 MATHEMATICAL FORMULATION FOR WING DESIGN METHOD - For any panelled baseline configuration, application of the wing design method requires that the arrays ϕ_i and $\partial\phi_i/\partial S_{k_S}$ have been calculated a priori. ϕ_i is the perturbation potential at the i_{th} control point and $\partial\phi_i/\partial S_{k_S}$ is the rate of change of ϕ_i with respect to independent geometry perturbation number k_S. As described previously MCAERO and DMCAERO will automatically calculate the required arrays and store them on a computer disk file. The wing design method can then be used to calculate the geometry perturbation that most nearly matches prescribed aerodynamic properties within the limitations of a minimal least square error.

At any panel center selected by the user, one or more properties can be prescribed. The property can be either pressure coefficient (C_p) or velocity component in an arbitrary, specified direction. The prescribed value of an aerodynamic quantity at a panel center is designated Q_{P_i}, where there is one index i for each prescribed value ($1 \le i \le N_{DES}$).

Arbitrary geometric constraints can be imposed upon the independent unknowns (S_{k_S}). Each geometric constraint is expressed as a linear equation

$$\sum_{i_D=1}^{N_D} \left[c_{i_D} \cdot S_{k_S(i_D)} \right] = CRHS \tag{1}$$

Where N_D, c_{i_D}, and CRHS are arbitrary values specified by the user. Depending upon the values specified, the constraint equation can be used to fix the cross-sectional area of a wing section, fix the thickness at one point, and so forth.

The aerodynamic design problem can now be expressed in mathematical form. The objective is to calculate the array of independent geometric unknowns S_{k_S} ($1 \leq k_S \leq NKS$) that will minimize the following function E.

$$\begin{aligned} E = & \sum_{i=1}^{N_{DES}} \left\{ WDES_i^2 \cdot AREA_i \cdot (Q_i - Q_{P_i})^2 \right\} \\ & + \sum_{j=1}^{N_{CON}} \left\{ WCON_j^2 \cdot \left[\sum_{i_D=1}^{n_D} c_{i_D} \cdot S_{k_S(i_D)} - CRHS \right]_j^2 \right\} \end{aligned} \tag{2}$$

where (1) each i ($1 \leq i \leq N_{DES}$) corresponds to one prescribed aerodynamic quantity at one panel center,

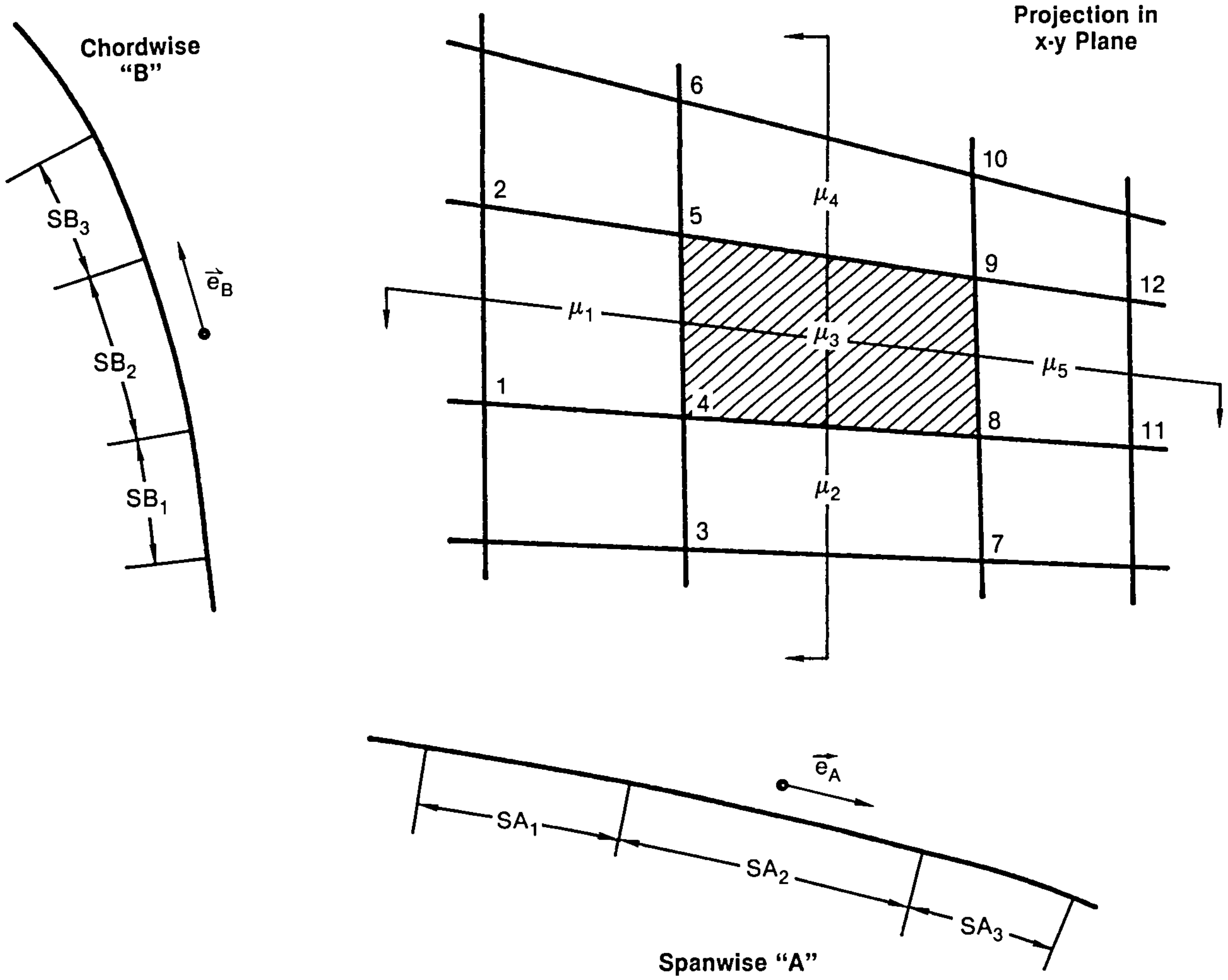
(2) each j ($1 \leq j \leq N_{CON}$) corresponds to a geometric constraint equation,

(3) the weights $WDES_i$ and $WCON_j$ are specified by the user (typically, $WDES_i = 1.00$ and $WCON_j \gg 1.00$),

(4) $AREA_i$ is the area on the baseline configuration of the panel corresponding to prescribed quantity Q_{p_i}, and

(5) ($Q_i - Q_{p_i}$) is the difference between the calculated and prescribed values of an aerodynamic property.

Iteration is used to solve for the array of unknowns S_{k_S} corresponding to a minimum value of E from equation 2. As shown in Figure 9, each iteration cycle is divided into an inverse step in which the geometry perturbation is calculated and a direct step in which the perturbed or updated geometry is analyzed. In order to reduce costs of the design method, the pressure distribution for the complete configuration is analyzed only for the zeroth (baseline) and final iteration cycle. For the intermediate cycles, the pressure distribution for only the design region is calculated using a procedure depicted in Figure 14. As illustrated, the spanwise and chordwise components of the perturbation velocity (designated q_A and q_B) are calculated using a one-dimensional interpolation through (μ_1, μ_3 and μ_5) and (μ_2, μ_3, and μ_4), respectively. Once q_A and q_B are known, the total velocity vector for the panel is determined as follows



$\{q_A; q_B\}$ is {Spanwise; Chordwise} Component of Perturbation Velocity and is Calculated by One-Dimensional Interpolation Through $\{(\mu_1, \mu_3, \mu_5); (\mu_2, \mu_3, \mu_4)\}$.

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Figure 14. Velocity Calculation on Design Panels

$$\vec{V} = \{\vec{V}_\infty + \sigma\vec{e}_N + q_B\, \vec{e}_B + q_c\, \vec{e}_c\} \tag{3}$$

where

σ is the panel source strength
$\vec{e}_N$ is the panel normal vector
q_c is defined as

$$q_c = \frac{q_A - q_B\, \vec{e}_B \cdot \vec{e}_A}{\vec{e}_c \cdot \vec{e}_A}$$

$\vec{e}_c$ is $\vec{e}_B \times \vec{e}_N$

In the inverse step of each iteration cycle, the matrix of derivatives is calculated. Then the change in Q_i induced by a small perturbation to the array S_{k_S} can be expressed as

$$dQ_i = \sum_{k_S=1}^{NKS} \frac{\partial Q_i}{\partial S_{k_S}} \cdot dS_{k_S} \tag{4}$$

By incorporating equation (4) in equation (2) and minimizing E with respect to dS_{k_S}, a system of linear, algebraic equations is established. The solution by standard matrix algebra provides the values dS_{k_S}. The updated geometry is then analyzed by the Perturbation Analysis Method.

One might expect that recalculation of the matrix $[\frac{\partial Q_i}{\partial S_{k_S}}]$ during each iteration cycle would require substantial computing expense. However, the following approach has proved to be both very efficient and accurate.

For each panel in the design region, an aerodynamic quantity Q_i, which can be either pressure coefficient or a velocity component at the panel center, is calculated. If Q_i is a pressure coefficient then

$$\begin{aligned} Q_i = C_p = & -2/V_{REF}^2\, [V_{\infty_x} V_x + V_{\infty_y} V_y + V_{\infty_z} V_z] \\ & - [\beta^2 V_x^2 + V_y^2 + V_z^2]/V_{REF}^2 \end{aligned} \tag{5}$$

$$\begin{aligned} \frac{\partial Q_i}{\partial S_{k_S}} = & -2/V_{REF}^2\, [V_{\infty_x} + \beta^2 V_x,\ V_{\infty_y} + V_y,\ V_{\infty_z} + V_z] \\ & \cdot \frac{\partial}{\partial S_{k_S}} [V_x,\ V_y,\ V_z] \end{aligned} \tag{6}$$

where

$$\beta^2 = 1 - M_\infty^2$$

V_x, V_y, V_z are the perturbation velocity components.

If Q_i is a velocity component in an arbitrary fixed direction (with direction cosines COSX, COSY, COSZ), then

$$Q_i = \vec{V} \cdot (\text{COSX, COSY, COSZ}) \tag{7}$$

$$\frac{\partial Q_i}{\partial S_{k_S}} = (\text{COSX, COSY, COSZ}) \cdot \frac{\partial}{\partial S_{k_S}} [V_x, V_y, V_z] \tag{8}$$

As shown by equations (6) and (8), the problem of calculating the derivatives of Q_i essentially reduces to calculating the velocity derivatives

$$\left(\frac{\partial V_x}{\partial S_{k_S}}, \frac{\partial V_y}{\partial S_{k_S}}, \frac{\partial V_z}{\partial S_{k_S}}\right)$$

At the center of the panel the velocity vector $\vec{V}$ can be represented as

$$\vec{V} = \vec{V} + \vec{\nabla}\phi \tag{9}$$

The gradient of the perturbation potential $\vec{\nabla}\phi$ can be calculated by numerical differentiation through 21 neighboring control points j. Mathematically this is

$$\vec{\nabla}\phi = \sum_{j=1}^{21} \phi_j \cdot (a_j\vec{e}_x + b_j\vec{e}_y + c_j\vec{e}_z) \tag{10}$$

where the scalars a_j, b_j and c_j are functions of the geometry at the panel corner points.

By substituting equation (10) into equation (9) and differentiating, the desired velocity derivatives can be expressed as

$$\frac{\partial}{\partial S_{k_S}} (V_x, V_y, V_z) = \sum_{j=1}^{21} \frac{\partial \phi_j}{\partial S_{k_S}} \cdot (a_j\vec{e}_x + b_j\vec{e}_y + c_j\vec{e}_z) + \phi_j \cdot \frac{\partial}{\partial S_{k_S}} (a_j\vec{e}_x + b_j\vec{e}_y + c_j\vec{e}_z) \tag{11}$$

The only term in equation (11) that requires substantial expense to compute is $\frac{\partial \phi j}{\partial S_{k_S}}$. Fortunately, it is also the only term that is nearly independent of perturbations to wing thickness, camber, and twist. Therefore, the precalculated baseline matrix $[\frac{\partial \phi i}{\partial S_{k_S}}]$ that is available on computer disk file from DMCAERO can be used in equation (11) during every iteration cycle. Substitution of equation (11) into equation (6) or (8) yields the desired value, $\frac{\partial Q i}{\partial S_{k_S}}$.

A significant feature of the preceding approach is that the accuracy of the calculated matrix $[\frac{\partial Q i}{\partial S_{k_S}}]$ is competitive with an exact first order expansion during each iteration cycle. However, much less computing effort is required.

In fact, the number of computations required for a complete iteration cycle is relatively small. The reason is apparent upon consideration of each calculation step in Figure 9. For example, consider the system of linear, algebraic equations to be solved for the perturbations to the independent unknowns. Typically, fewer than one hundred unknowns are sufficient for wing design, compared to several hundred for a conventional panel method analysis of a wing-fuselage. Also, consider the last step of each iteration cycle - analyzing the updated geometry. The extremely efficient Perturbation Analysis Method is used for that calculation. This coupled with the fact that for the intermediate cycles only the design region is analyzed results in a very efficient method for designing wing section geometry.

The wing design method has been automated and is fully operational on the McDonnell Douglas CYBER 176. The production computer program is designated "DESIGN". A demonstration of the accuracy, efficiency, and numerical stability of the method is presented in the next section.

3.3 EXAMPLE DESIGN SOLUTIONS - Two example solutions by Program DESIGN are presented below. The calculations were performed on the McDonnell Douglas CDC CYBER 176.

In the first example, the objective is to design a fighter-type wing section geometry by starting from the baseline NACA-0012 wing panelling of Figure 15. The fighter wing pressure distribution at 0° angle of attack was prescribed at the center of each of the 208 panels. The converged solution after 3 iterations is presented in Figure 15. The relative cost of the designed solution is shown in Figure 8 where the design solution costs approximately 1/4 that of the conventional MCAERO solution.

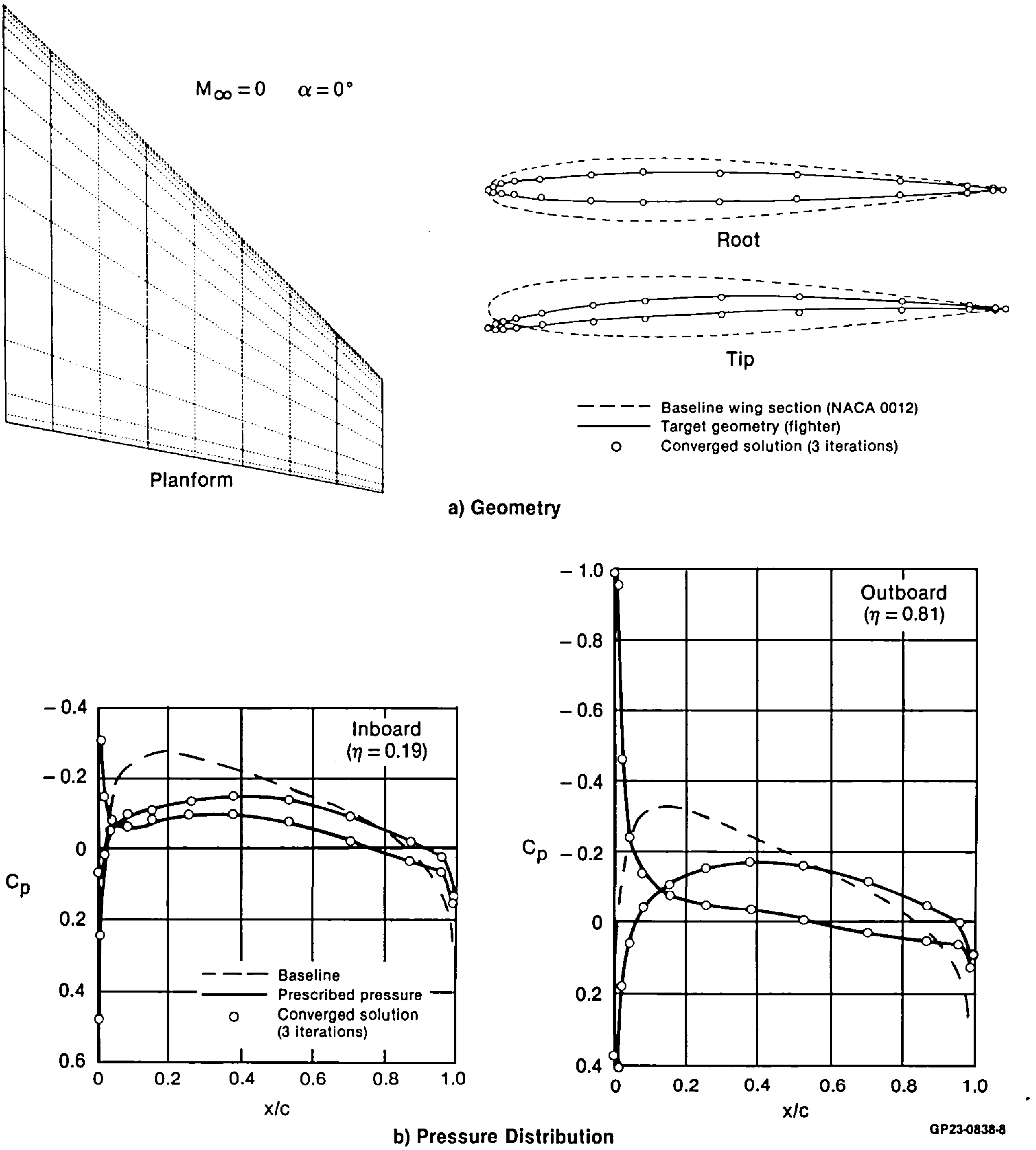


Figure 15. Example Wing Design Solution

In the second example, the objective is to design a wing on a fuselage-nacelle-tail configuration. The baseline is shown in Figure 16. For this case, the prescribed pressure was basically a linear pressure distribution which removed the leading edge pressure spike. The resulting pressure and geometry are shown in Figure 17. Although the solution converged in two iterations, a third iteration was calculated for verification. The computing cost of the design solution was only about 1/25 that of the conventional solution (see Figure 8). Based upon these and other test cases, it was concluded that the 3-D Wing Design Method is indeed an accurate, efficient and cost effective method of designing wing section geometries corresponding to prescribed pressure distributions. The next section describes the interactive graphics module.

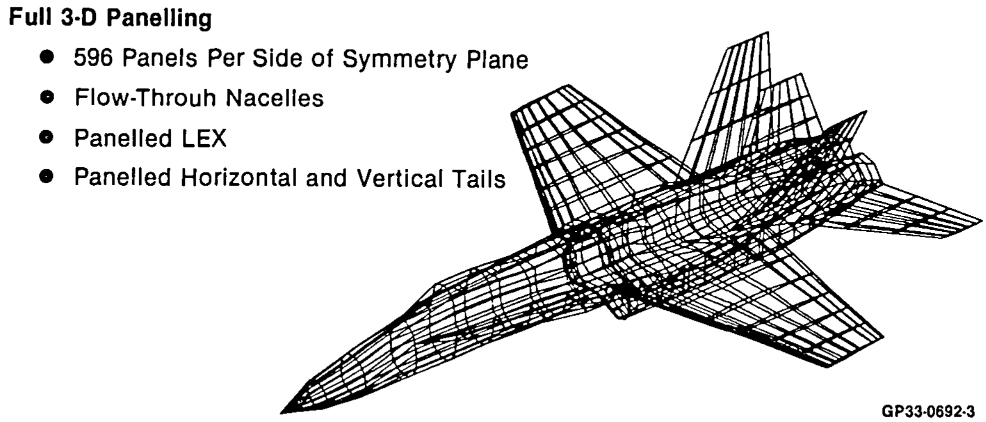


Figure 16. Isometric of Panelled F-18

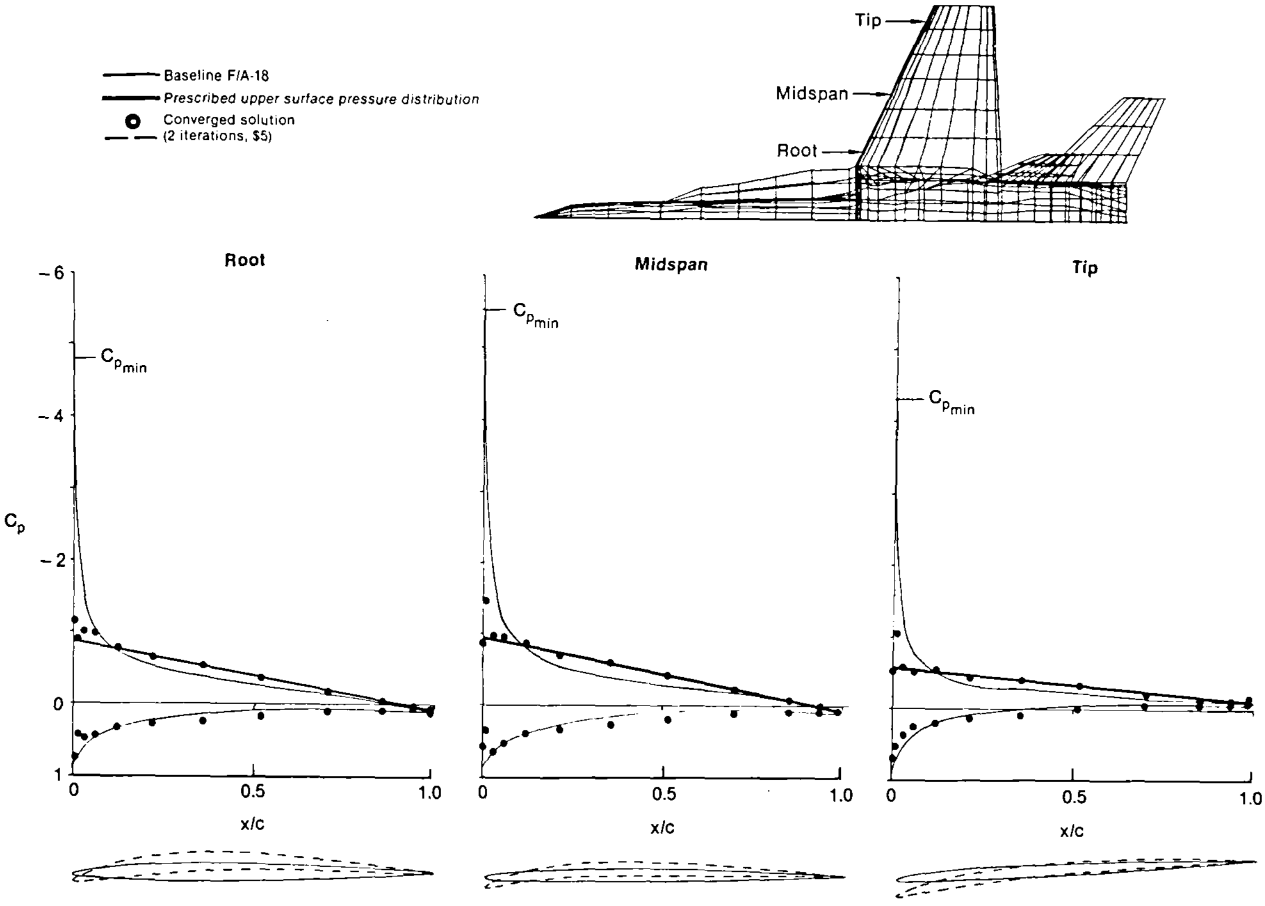


Figure 17. MCAERO Wing Design Test Case
F/A-18 Low Speed $\alpha = 8°$

GP33-0026-11

4. INTERACTIVE GRAPHICS OPTION

The interactive graphics option is a stand-alone self-documented code which takes an output from the 3-D wing design code, plots user selected pressure and geometries, and if desired, creates a new input file, interactively, for the design code. This graphics module has 4 options built into it. The first allows the user to plot any or all the geometry segments in a three view plot (Figure 18). The second option allows the user to plot a wing section geometry and the corresponding pressure distribution (Figure 19).

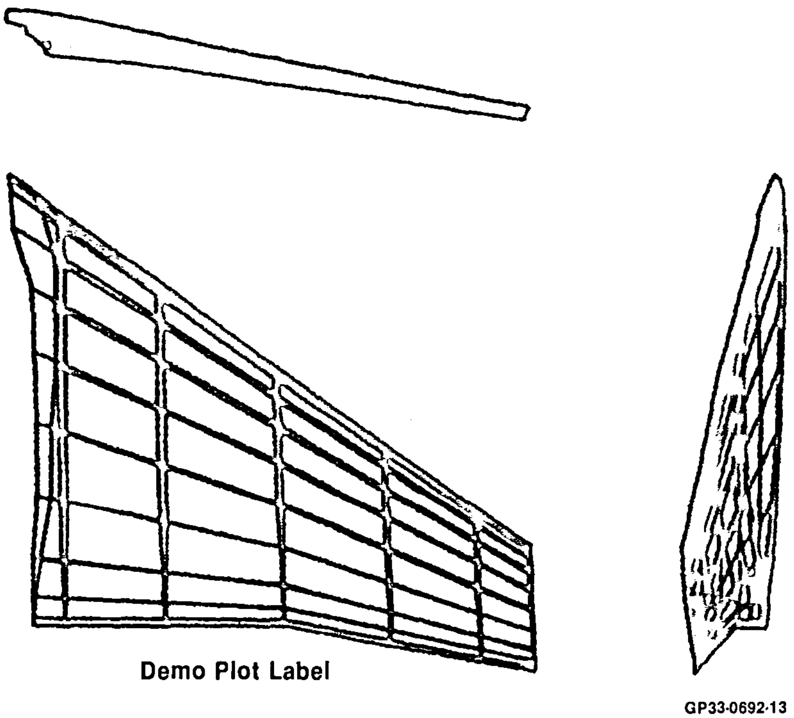


Figure 18. Three View Geometry
Interactive Graphics

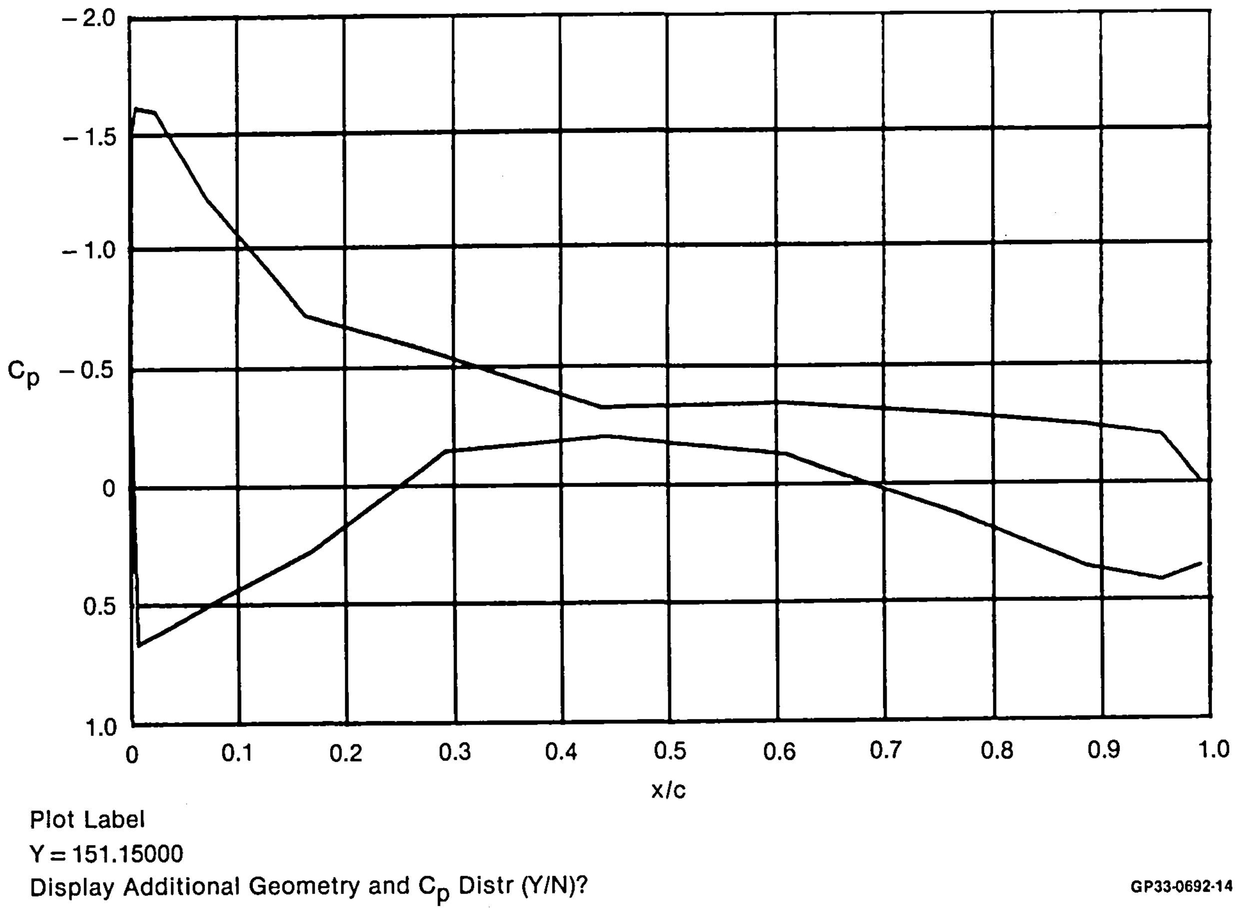
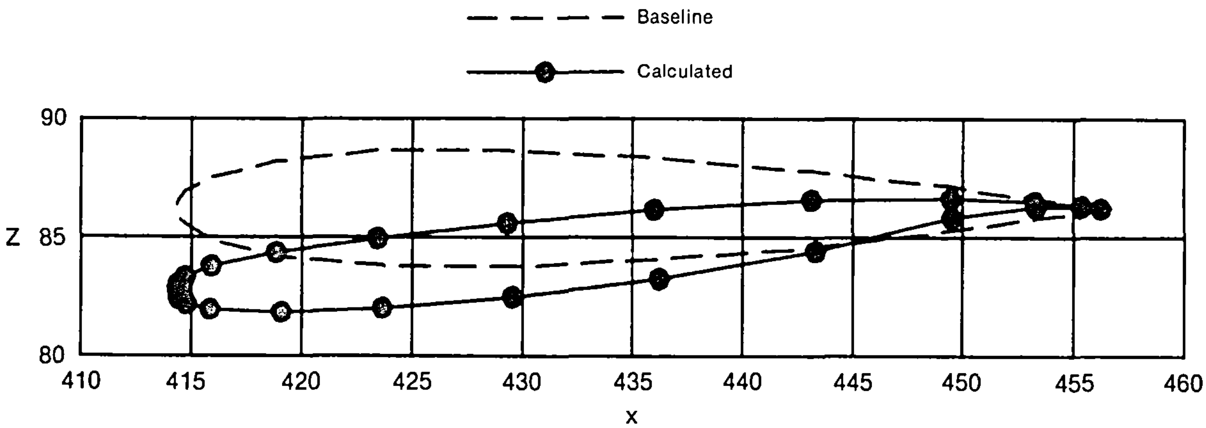


Figure 19. Wing Section and Pressure Distribution

Interactive Graphics

Options 3 and 4 are used when the user wants to modify either the initial wing geometry or prescribed pressure distribution for another design case. Option 3 allows the user to modify the initial guess of the starting geometry. The code displays the selected wing section, both initial baseline and the resulting designed wing (Figure 20). If the user requests geometric changes, then the point numbers are placed on the points that the user may change (Figure 21), the screen is then cleared, and the point number and value are displayed (Figure 22). The user may then select new values. When done, the code displays the new geometry (Figure 23).

Change Next Iteration Geometry? (Enter Y or N)



Plot Label
Wing Section = ?
Y = 181.40000

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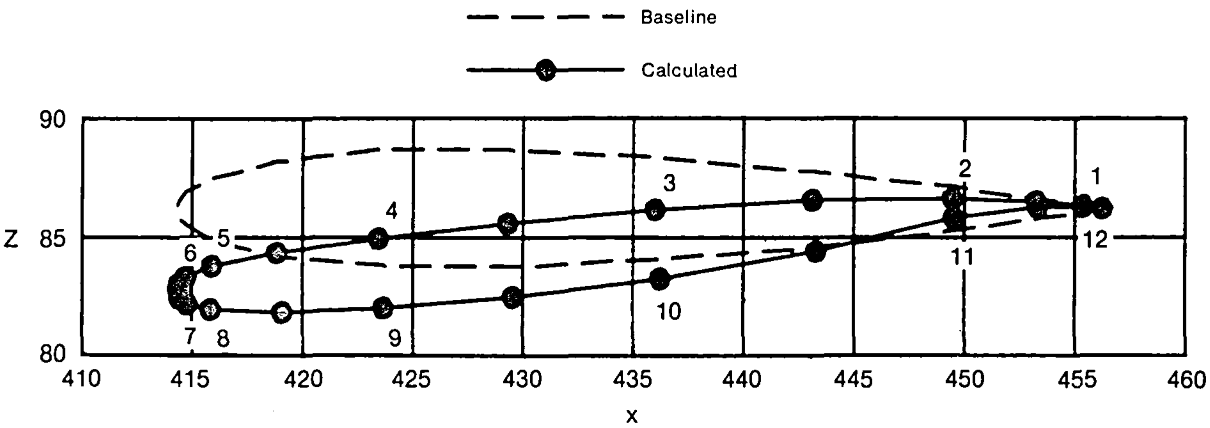
Figure 20. Wing Section Geometry
Interactive Graphics

Change Next Iteration Geometry? (Enter Y or N)
?Y
Type "C" to Continue
?C



Plot Label
Wing Section = ?
Y = 181.40000

GP33-0692-16

Figure 21. Wing Section Geometry
Interactive Graphics

```
POINT #          Z VALUE
     1           86.34600
     2           86.66900
     3           86.15800
     4           84.95500
     5           83.74300
     6           82.97300
     7           82.38900
     8           81.96400
     9           81.98400
    10           83.22100
    11           85.79700
    12           86.30700
 ENTER POINT NO.,NEW Z VALUE
 ENTER 0,0 TO END
?1,87
?2,87
?11,87
?12,87
?0,0
 DISPLAY GRID?
?y
```

GP33-0692-17

Figure 22. Wing Geometric Points
Interactive Graphics

Accept Changes to This Section (Enter Y or N)
?Y

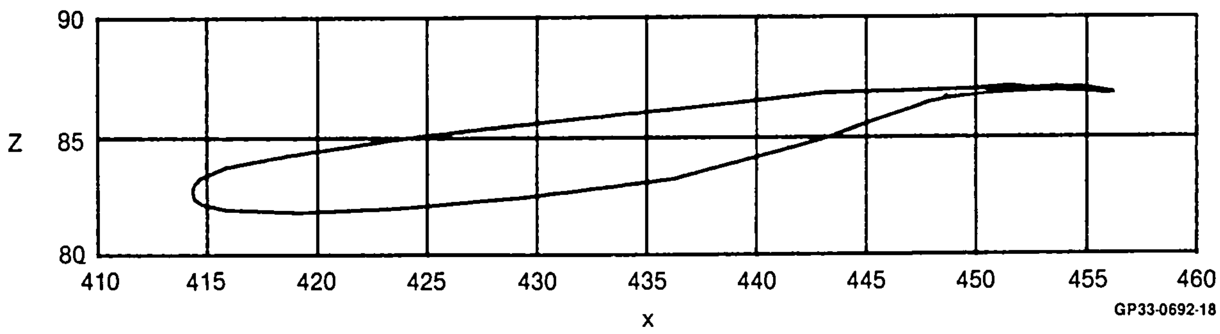


Figure 23. Updated Wing Section Geometry
Interactive Graphics

By using Option 4, the user may change the prescribed pressure distribution. Initially the code displays both the calculated and prescribed pressure at the requested station (Figure 24). If the user wants to change any pressures, the pressure points are numbered (Figure 25), screen is cleared, and the point numbers and pressures are displayed (Figure 26). The user may then select the desired points and their new values. When done, the code displays the new prescribed pressure as shown in Figure 27. Once the user is satisfied with the new pressure and geometry, the code generates an input file for the wing design code. This iterative procedure can be repeated again until a final desired solution is obtained.

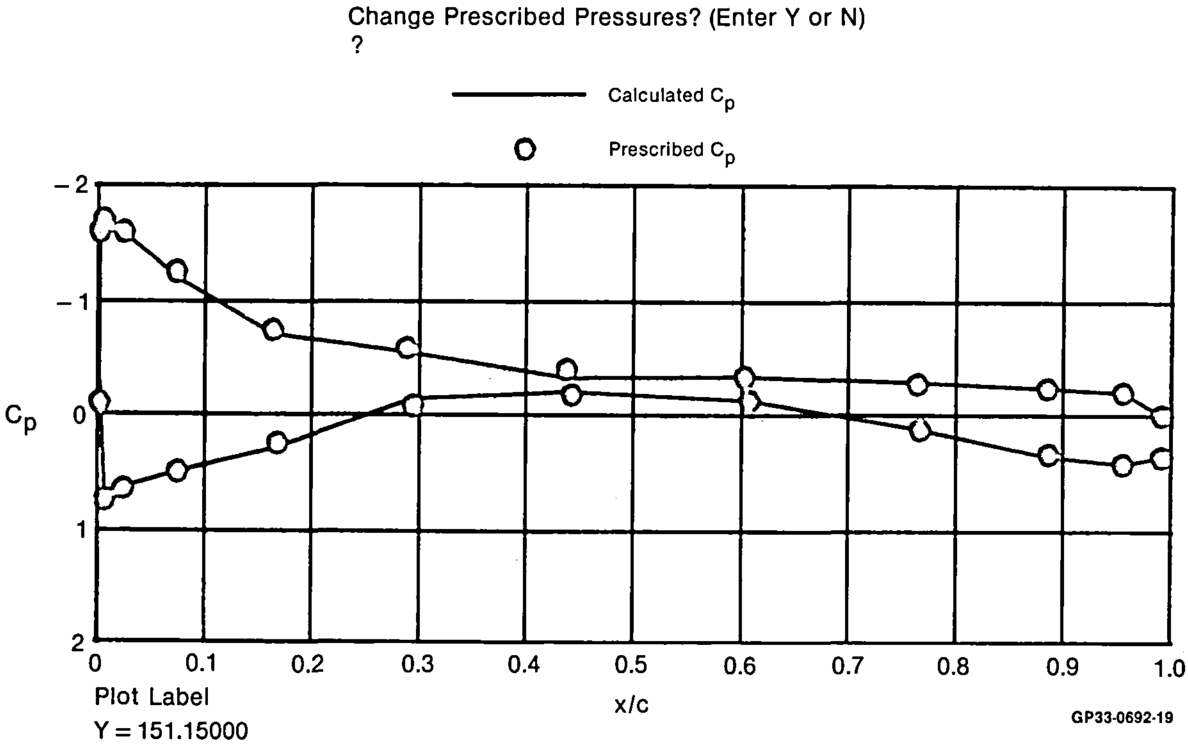
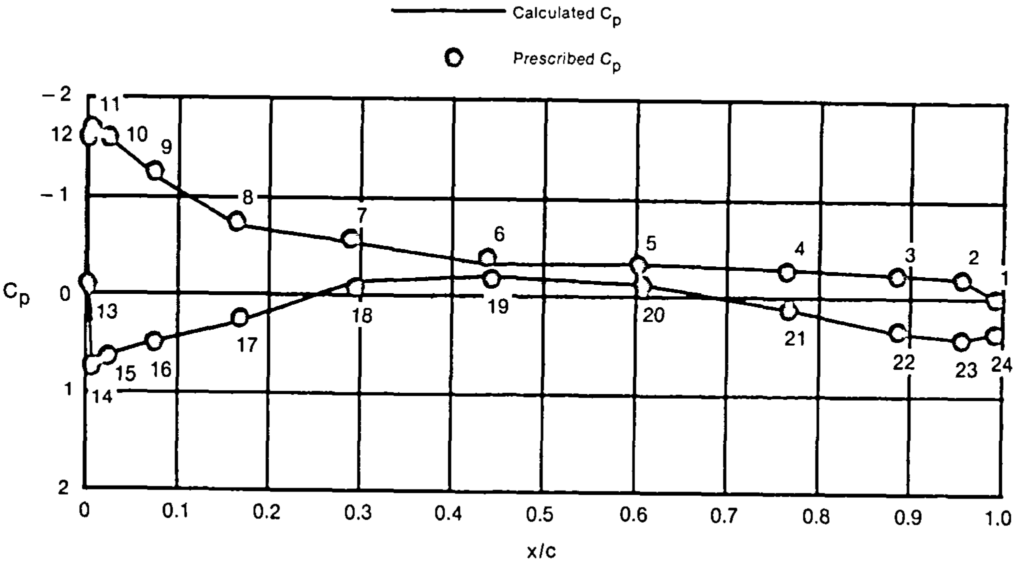


Figure 24. Wing Section Pressure Distribution

Interactive Graphics

Change Prescribed Pressures? (Enter Y or N)
?Y
Type "C" to Continue
?



Plot Label
Y = 151.15000

GP33-0692-20

Figure 25. Wing Section Pressure Distribution
Interactive Graphics

POINT #	PRESSURE
1	0.00000
2	-.20000
3	-.23000
4	-.27000
5	-.33000
6	-.40000
7	-.59000
8	-.75000
9	-1.25000
10	-1.60000
11	-1.70000
12	-1.60000
13	-.10000
14	.75000
15	.64000
16	.50000
17	.25000
18	-.08000
19	-.18000
20	-.12000
21	.13000
22	.33000
23	.42000
24	.36000

```
 ENTER POINT NO.,NEW CP VALUE
 ENTER 0,0 TO END
?6,.45
?7,.65
?8,.90
?0,0
```

GP33-0692-21

Figure 26. Wing Pressures
Interactive Graphics

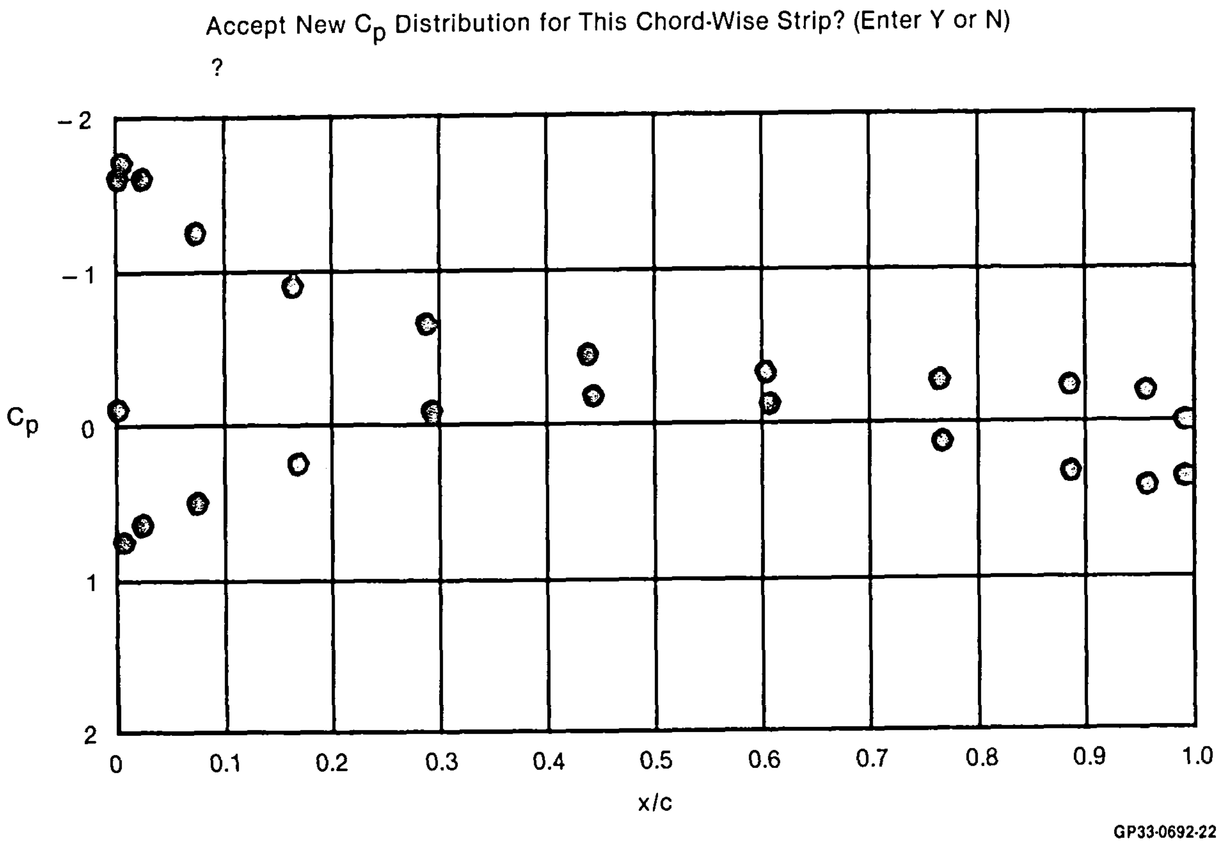


Figure 27. Modified Wing Pressure Distribution
Interactive Graphics

5. CONCLUSIONS

The perturbation analysis and wing design methods are similar to classical thin wing theory in the sense that small disturbance "linearized" assumptions are employed. This mathematical simplification generates extensive computational savings for aerodynamic problems involving successive iteration, such as design.

On the other hand, the restrictions of classical thin wing theory have been eliminated. Compared to an exact potential flow solution, the present approach is quite accurate for thick wings, large leading edge radius or camber, and high angle of attack. The success of the wing design method is attributed to the inclusion of all significant first-order geometry-pressure perturbation terms in each iteration cycle. This leads to rapid solution convergence, in spite of the fact that the entire distribution of surface potential is constructed by simple linear extrapolation. With the inclusion of the interactive graphics module, the wing design method has become a unique, powerful, highly efficient method for designing wing geometries which correspond to prescribed pressure distributions.

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16. Abstract
A reliable and efficient iterative method has been developed for designing wing section contours corresponding to a prescribed subcritical pressure distribution. The design process is initialized by using MCAERO (MCAIR 3-D Subsonic Potential Flow Analysis Code) to analyze a baseline configuration. A second program DMCAERO is then used to calculate a matrix containing the partial derivative of potential at each control point with respect to each unknown geometry parameter by applying a first-order expansion to the baseline equations in MCAERO. This matrix is calculated only once but is used in each iteration cycle to calculate the geometry perturbation and to analyze the perturbed geometry. The potential on the new geometry is calculated by linear extrapolation from the baseline solution. This extrapolated potential is converted to velocity by numerical differentiation, and velocity is converted to pressure by using Bernoulli's equation. There is an interactive graphics option which allows the user to graphically display the results of the design process and to interactively change either the geometry or the prescribed pressure distribution. Not only is this design procedure accurate for large perturbations, but the cost of each iteration cycle is more than two orders of magnitude less than a conventional analysis solution. Examples of the design process are presented to demonstrate that the method is accurate, numerically stable, and converges in only three to five iterations.

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